U.S. DEPARTMENT OF COMMERCE
National Technical Information Service

AD-A024 541

COMBUSTOR DESIGN FOR LOW COST EXPENDABLE TURBOJETS

AIR FORCE INSTITUTE OF TECHNOLOGY

MARCH 1976

KEEP UP TO DATE

Between the time you ordered this report—which is only one of the hundreds of thousands in the NTIS information collection available to you—and the time you are reading this message, several *new* reports relevant to your interests probably have entered the collection.

Subscribe to the **Weekly Government Abstracts** series that will bring you summaries of new reports *as soon as they are received by NTIS* from the originators of the research. The WGA's are an NTIS weekly newsletter service covering the most recent research findings in 25 areas of industrial, technological, and sociological interest—invaluable information for executives and professionals who must keep up to date.

The executive and professional information service provided by NTIS in the **Weekly Government Abstracts** newsletters will give you thorough and comprehensive coverage of government-conducted or sponsored research activities. And you'll get this important information within two weeks of the time it's released by originating agencies.

WGA newsletters are computer produced and electronically photocomposed to slash the time gap between the release of a report and its availability. You can learn about technical innovations immediately—and use them in the most meaningful and productive ways possible for your organization. Please request NTIS-PR-205/PCW for more information.

The weekly newsletter series will keep you current. But *learn what you have missed in the past* by ordering a computer **NTISearch** of all the research reports in your area of interest, dating as far back as 1964, if you wish. Please request NTIS-PR-186/PCN for more information.

WRITE: Managing Editor
5285 Port Royal Road
Springfield, VA 22161

Keep Up To Date With SRIM

SRIM (Selected Research in Microfiche) provides you with regular, automatic distribution of the complete texts of NTIS research reports *only* in the subject areas you select. SRIM covers almost all Government research reports by subject area and/or the originating Federal or local government agency. You may subscribe by any category or subcategory of our WGA (**Weekly Government Abstracts**) or **Government Reports Announcements and Index** categories, or to the reports issued by a particular agency such as the Department of Defense, Federal Energy Administration, or Environmental Protection Agency. Other options that will give you greater selectivity are available on request.

The cost of SRIM service is only 45¢ domestic (60¢ foreign) for each complete microfiched report. Your SRIM service begins as soon as your order is received and processed and you will receive biweekly shipments thereafter. If you wish, your service will be backdated to furnish you microfiche of reports issued earlier.

Because of contractual arrangements with several Special Technology Groups, not all NTIS reports are distributed in the SRIM program. You will receive a notice in your microfiche shipments identifying the exceptionally priced reports not available through SRIM.

A deposit account with NTIS is required before this service can be initiated. If you have specific questions concerning this service, please call (703) 451-1558, or write NTIS, attention SRIM Product Manager.

This information product distributed by

NTIS U.S. DEPARTMENT OF COMMERCE
National Technical Information Service
5285 Port Royal Road
Springfield, Virginia 22161

COMBUSTOR DESIGN FOR
LOW COST
EXPENDABLE TURBOJETS

THESIS

Raymond L. Greene
Major USAF

GAE/AE/76M-1

AD-A024325

COMBUSTOR DESIGN FOR
LOW COST
EXPENDABLE TURBOJETS

THESIS

Raymond L. Greene
Major USAF

GAE/AE/76M-1

Approved for public release; distribution unlimited

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

1. REPORT NUMBER GAE/AE/76M-1	2. GOVT ACCESSION NO.
3. RECIPIENT'S CATALOG NUMBER	
4. TITLE (and Subtitle) COMBUSTOR DESIGN FOR LOW-COST EXPENDABLE JET ENGINES	5. TYPE OF REPORT & PERIOD COVERED MS Thesis
	6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s) Raymond L. Greene Major USAF	8. CONTRACT OR GRANT NUMBER(s)
9. PERFORMING ORGANIZATION NAME AND ADDRESS Air Force Institute of Technology (AFIT-EN) Wright-Patterson AFB, Ohio 45433	10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS
11. CONTROLLING OFFICE NAME AND ADDRESS	12. REPORT DATE March, 1976
	13. NUMBER OF PAGES 53
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)	15. SECURITY CLASS. (of this report) Unclassified
	15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

Approved for public release; IAW AFR 190-17

JERRY C. HIX, Capt, USAF
Director of Information

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Small jet engine, Low-cost jet engine
Combustor, Burner
Turbocharger, Turbo-supercharger
RPV engine

...CES SUBJECT TO CH...

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The development of two experimental burners was completed as part of a project to evaluate the suitability of the use of converted reciprocating engine turbochargers as expendable jet engines for remotely piloted vehicles. As a result of analytic studies, Burner "A" was constructed as a reverse flow burner and was designed to accept a mass flow up to 0.75 lbm/sec. Burner "B" was also reverse and designed for a mass flow up to 1.5 lbm/sec. Both burners operated successfully on their respective turbochargers.

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

Maximum thrust achieved was 55 pounds. Analytic results indicated that this value could be extended to 65-70pounds without afterburning and to 95-100 pounds with afterburning.

COMBUSTOR DESIGN FOR

LOW COST

EXPENDABLE TURBOJETS

THESIS

Presented to the Faculty of the School of Engineering of

the Air Force Institute of Technology

Air University

in Partial Fulfillment of the

Requirements for the Degree of

Master of Science

by

Raymond L. Greene, B.S.,E.S.

Major USAF

Graduate Aeronautical Engineering

March 1976

Approved for public release; distribution unlimited.

Preface

The increasing sophisication of defensive weapons during the latter part of the 1960's resulted in reduced effectiveness of direct action by manned aircraft. During this period, the success of short range guided weapons, such as laser guided bombs, anti-radiation missiles and other homing or guided weapons has been well documented. Recent developments in defensive systems have increased the need for advanced stand-off weapons and for remotely-piloted reconnaissance and decoy vehicles. Because of the high cost of small jet engines, these remotely-piloted vehicles have been relatively expensive.

The purpose of this study was to evaluate one alternative to the current high cost engine. This alternative was a proposal by the AF Aero Propulsion Laboratory to examine the suitability of using reciprocating engine turbo-superchargers as expendable small turbojet engines.

Specifically, this study covers the major component required for such a conversion, the combustor and its associated hardware.

I wish to thank Dr. William Elrod for the direction and support given to me in his capacity as my thesis advisor. I would also like to thank Mr. David Wilkinson from the Aero-Propulsion Laboratory, Mr. Millard Wolfe and the personnel of the AFIT Shop, Mr. William Baker, Mr. John Parks, and Mr. John Flahive for invaluable assistance and advice.

Raymond L. Greene

Contents

List of Figures

List of Tables

List of Symbols

Symbol	Quantity	Units
A	Area	sq. in.
$F(M)$	Function of Mach Number	- -
M	Mach number	- -
$\dot{m}$	Mass flow	lbm/sec
P	Pressure	psia
R	Gas constant	lbf-ft/^{o}R-lbm
Re	Reynolds number	- -
T	Temperature	^{o}R
u	Velocity	fps
W	Weight flow in combustor	lbm/sec
β	Orifice diameter ratio	- -
γ	Ratio of specific heats	- -

Subscripts

a	Air
f	Fuel
s	Static conditions
t	Total conditions

Abstract

The development of two experimental burners was completed as a part of a project to evaluate the suitability of the use of converted reciprocating engine turbochargers as expendable jet engines for remotely piloted vehicles. Analysis of the variables affecting the turbocharger engine indicated that a 10 percent variation in pressure loss in the burner would have a negligible effect on the thrust produced. A 10 percent variation in temperature at the burner exit was a significant factor.

Considering the analytic results, it was decided to construct two burners, both reverse flow to reduce the size of the overall machine. Burner "A" was designed for an air capacity of up to 0.75 lbm/sec with the exit leading directly into the turbine. The airflow through the inlet of the burner entered the annular plenum in a direction perpendicular to the burner axis. Total pressure losses in this burner were measured at 1 percent for cold flow and 3 percent for hot flow.

Burner "B" was designed increasing the velocity through the liner air holes and reducing the velocity in the combustion region. Because of these changes, pressure loss was increased to 8 percent. The air inlet was also changed to inject the air tangentially in order to introduce swirl into the burning region. The swirl created a more even exit temperature. This burner produced satisfactory results on the test rig and was further tested on the turbochargers.

Tests on the turbochargers resulted in pulsating and intermittant operation. Fuel nozzles with better atomization corrected this condition and allowed smooth operation of the machines with various turbine nozzles.

Final experimental results indicated the following factors had the greatest effect on the turbocharger-engine performance. In order of decreasing importance, these factors are (1) burner instability caused by burner/turbocharger interaction such as downstream torching of unburned fuel etc., (2) burner massflow limits which appeared to be lower than the turbocharger maximum flow, and (3) the airflow conditions in the burner plenum.

COMBUSTOR DESIGN FOR
LOW-COST
EXPENDABLE TURBOJETS

I. Introduction

The increasing sophistication of anti-aircraft defensive systems in recent years has created a need for expendable, low-cost vehicles to perform a variety of missions. The high cost of the current small jet engines has prevented the development of a true low-cost vehicle. Ajet engine built with a turbocharger is a possible solution to the problem of high engine cost. This study examines the combustor, a major component required for the development of low-cost engines. Concurrently, a study examining turbocharger performance was conducted by Capt. Tommy J. Kent (Ref. 5).

Background Information

The USAF has established a requirement for remotely-piloted vehicles (RPV's) to perform a variety of missions. These missions include tactical reconnaisance, electronic jamming and interrogation, ordnance delivery, decoy and target missions. Current engines which are available to meet these mission requirements are limited to small reciprocating engines and a few small turbojets which cost approximately $85 / lb thrust (Ref. 7). This limitation in powerplants has restricted the sizes of RPV's and made the jet-powered versions relatively expensive. In order to reduce the powerplant limitation, the AF Aero Propulsion

Laboratory is exploring alternative systems. A proposal for one of these systems is to convert an ordinary supercharger into a small turbojet engine by adding a combustor, inlet and nozzle. Superchargers are available in a wide range of sizes for approximately $600 each. The performance required for this application is 100 lb. static thrust and a one time operation for 30 minutes.

Objectives and Criteria

The purpose of this study is to design and test two combustion chambers for use in examining the feasibility of using converted turbochargers as low-cost expendable jet engines. Further, it is desired to identify and define burner characteristics which have the greatest effect on turbocharger performance. Suitable burner operation is considered to exist when continuous burning throughout the required range of turbocharger operation is achieved with no hot spots of sufficient intensity to cause physical damage to the machinery. For this study, no attempt was made to reduce weight. Figures 1 and 2 on the following pages show both turbochargers with the respective burners attached. The burner liners are shown with each machine.

Scope

The immediate problem was to provide a preliminary combustor with which testing of the turbocharger could begin. It was decided to design two combustors, a boiler plate model,

Burner "A", which could be used for immediate testing of the first engine (#J1) to be built using the Rajay turbocharger, and a more refined Burner "B" as a follow-on to be used on engine J2 to be built with the larger AiResearch turbocharger. Only minor modifications were to be made to Burner "A" while major changes were to be incorporated, as necessary, into Burner "B".

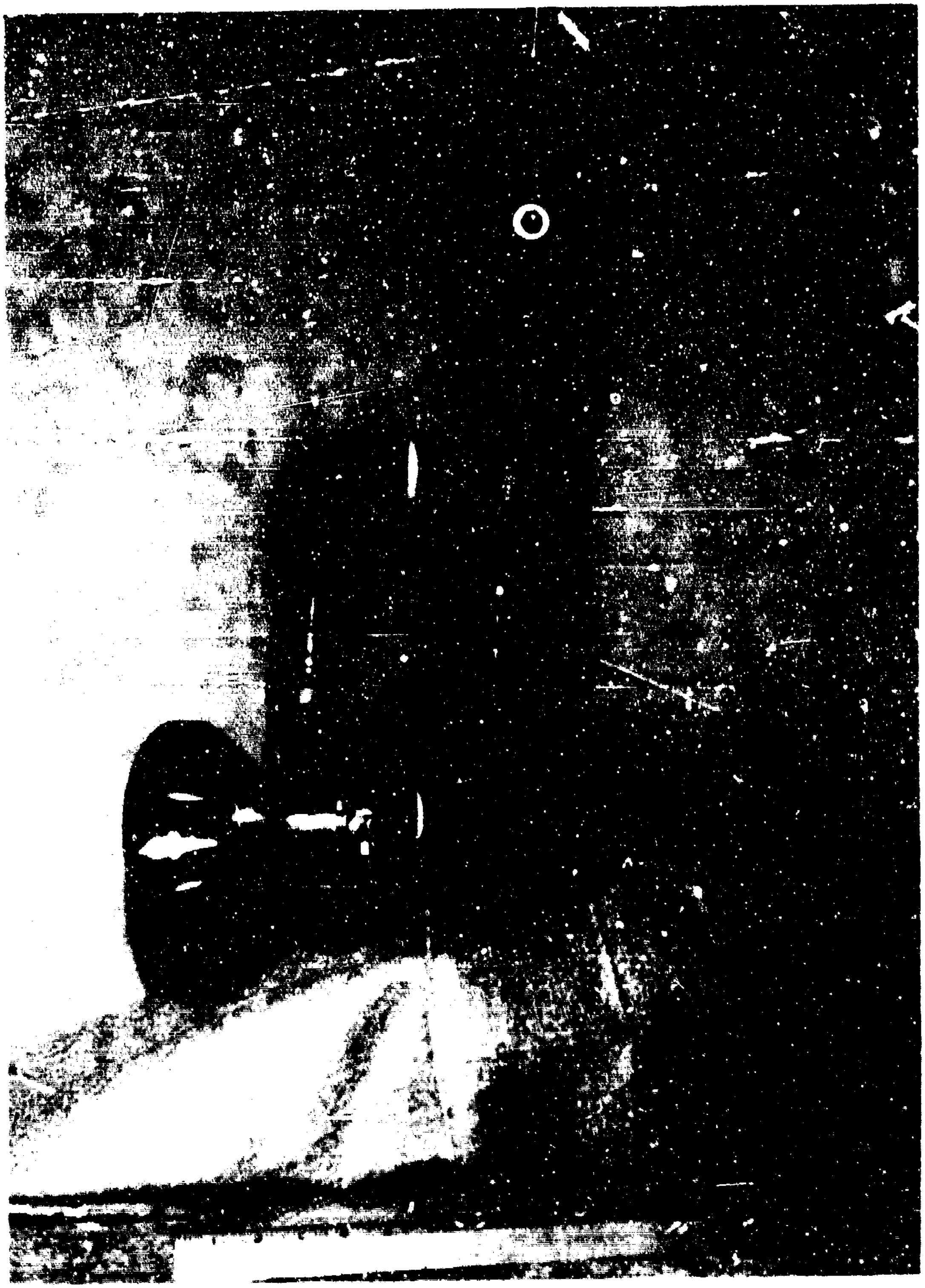


Fig. 1 Engine J-1 with Burner "A"

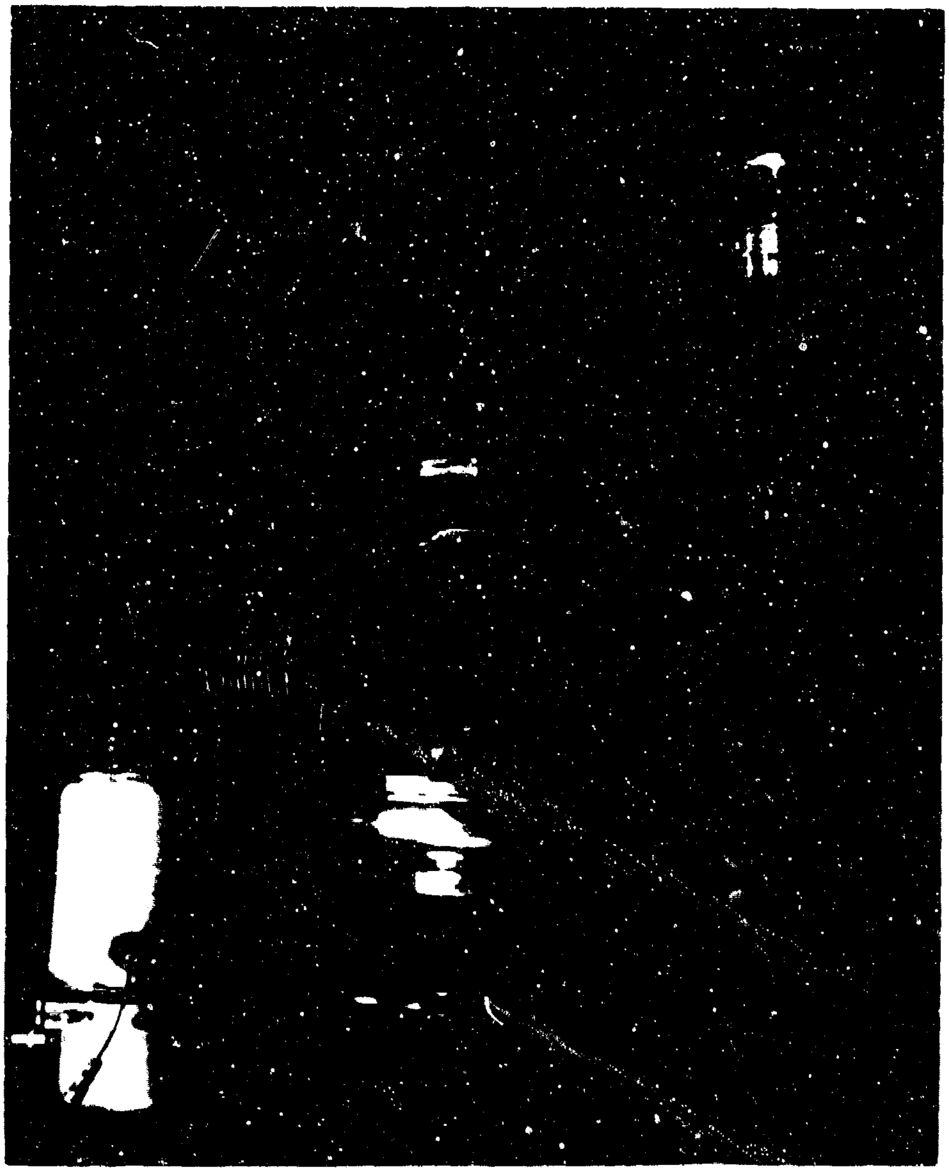


Fig. 2 Engine J-2 with Burner "B"

II. Description of the Combustor Designs

Two complete combustors were designed. Burner "A" for the smaller Rajay turbocharger (mass flow to .75 lbm/sec) and Burner "B" for the AiResearch turbocharger (mass flow to 1.5 lbm/sec). Each burner consisted of an outer shell forming a reverse flow plenum, a burner liner, adaptors at the inlet and exit to connect to the turbocharger, ignitors and the fuel spray nozzles.

Burner "A"

The outer shell of Burner "A" was constructed from steel pipe with a 3.5-inch inside diameter and a 13-inch length. The annular area around the liner was constructed equal to the cross-sectional area of the liner. The total air flow area through the liner from the annular area was three times the cross-sectional area of the liner in order to minimize pressure losses (See Appendix B). Air distribution was designed for 20% at the front of the liner for fuel atomization, 15% immediately behind the ignitors to complete stoichiometric conditions and 65% evenly distributed along the remaining liner length for cooling. The liner was made of 321 stainless steel and was 2.375 inches across the inside diameter. A swirler collar was located at the front of the liner. An adaptor and instrumentation collar, located at the end of the burner, served as a heat expansion joint. Figure 3 shows the general dimensions.

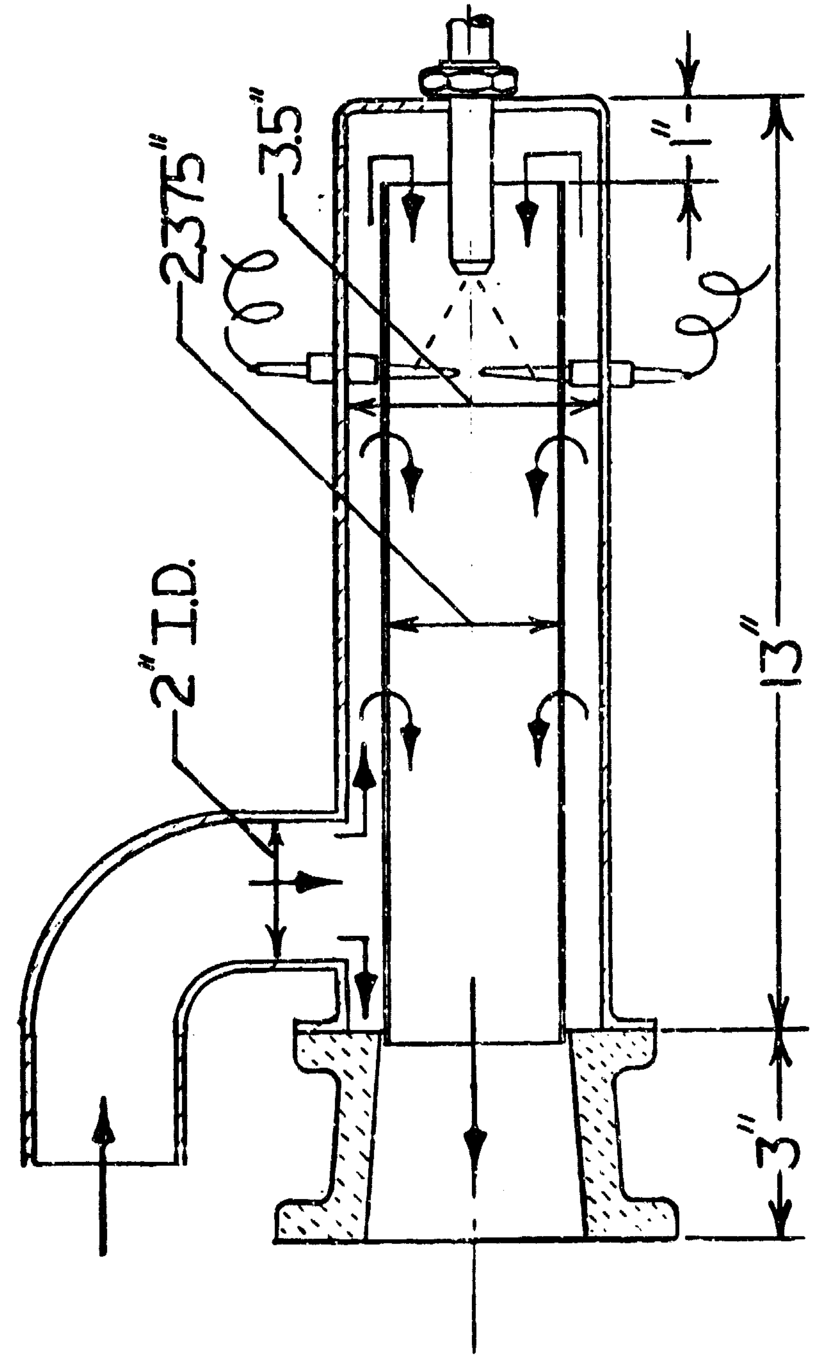


Fig. 3 - General Dimensions - Burner "A"

Burner "B"

The outer shell of Burner "B" was constructed with stainless steel sheet 1/16 inch thick. The forward dome, liner and ignitor were standard fittings from an MA-1A Jet Aircraft Starting Unit (JASU). An MA-1A liner was adapted for use in this burner because experimental results indicated that the velocity in Burner "A" was too high. Calculations showed the velocity in the stoichiometric section of Burner "B" to be approximately 30 ft/sec compared to 50 ft/sec in Burner "A". Inlet air flow was directed tangentially into the plenum. Overall length excluding the burner to turbine adaptor was 14 inches and overall diameter was $7\frac{1}{4}$ inches. Figure 4 shows the general dimensions.

Ignitors

For Burner "A" the ignitor electrodes were constructed of stainless steel and set opposite each other with a .25 inch gap between them centered inside the liner. Automotive sparkplugs were also used and proved satisfactory although starting was slower. An aircraft ignitor plug was used on Burner "B".

Fuel Nozzles

Oil furnace fuel nozzles, made by the Delavan Company, Des Moines, Iowa, were used in both combustors.

The fuel nozzles used had a solid cone spray at an 80 degree angle of spray and were rated at 4 gal/hr, 6 gal/hr, 8 gal/hr and 14 gal/hr. Fuel was delivered to the nozzles at pressures between 20 psi and 150 psi. The minimum pressure for atomization of the fuel was 15 - 20 psi. The nozzle location was

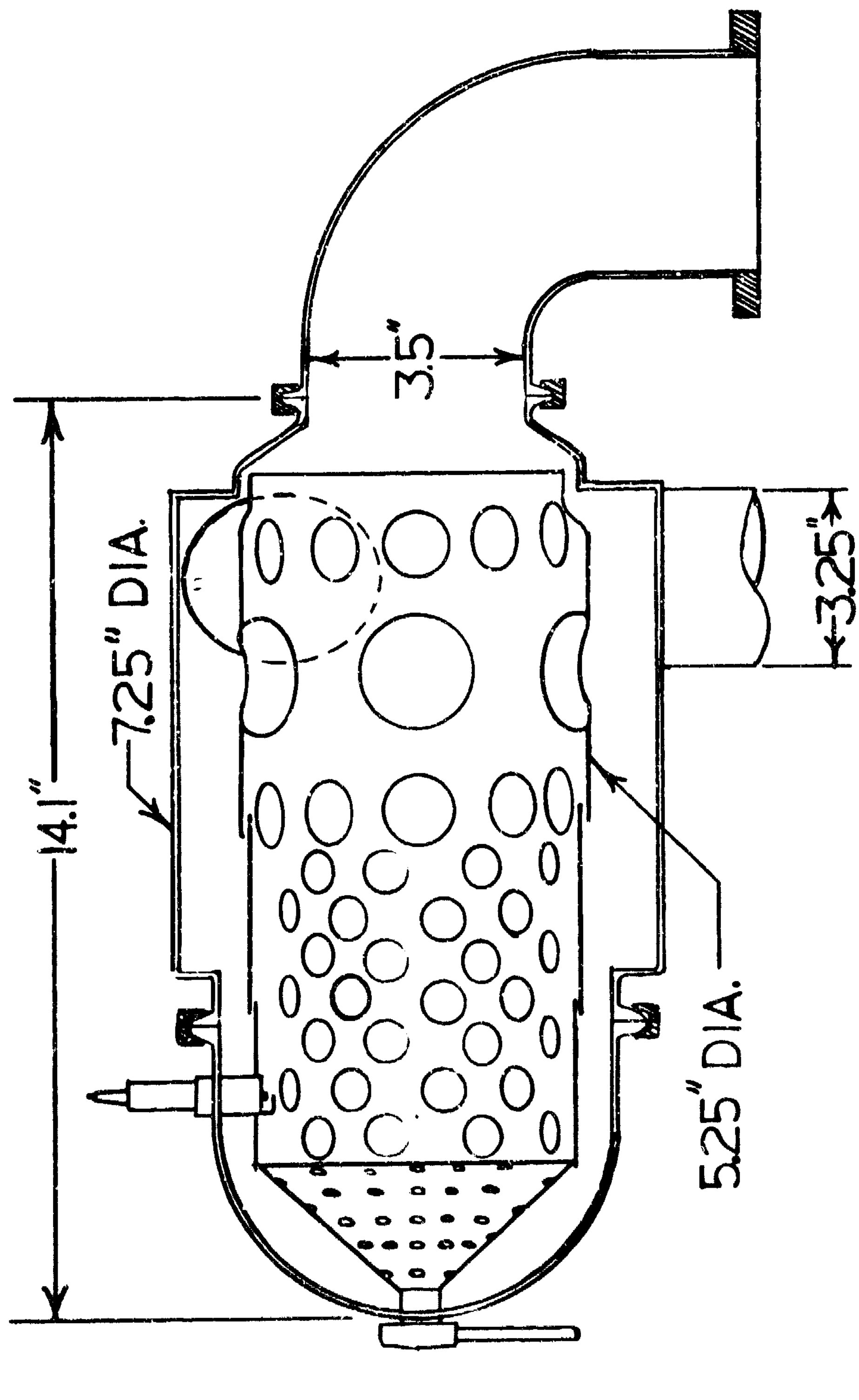


Fig. 4 - General Dimensions Burner "B"

adjustable along the burner axis in both combustors.

In addition to the furnace nozzles, a hollow cone spray nozzle from an MA-1A JASU was used on Burner "B". Minimum pressure for good atomization with this nozzle was 10 psi and flow rate changes were greater with fuel pressure changes. Figure 5 shows a comparison of flow rates for each nozzle over the operating pressure range.

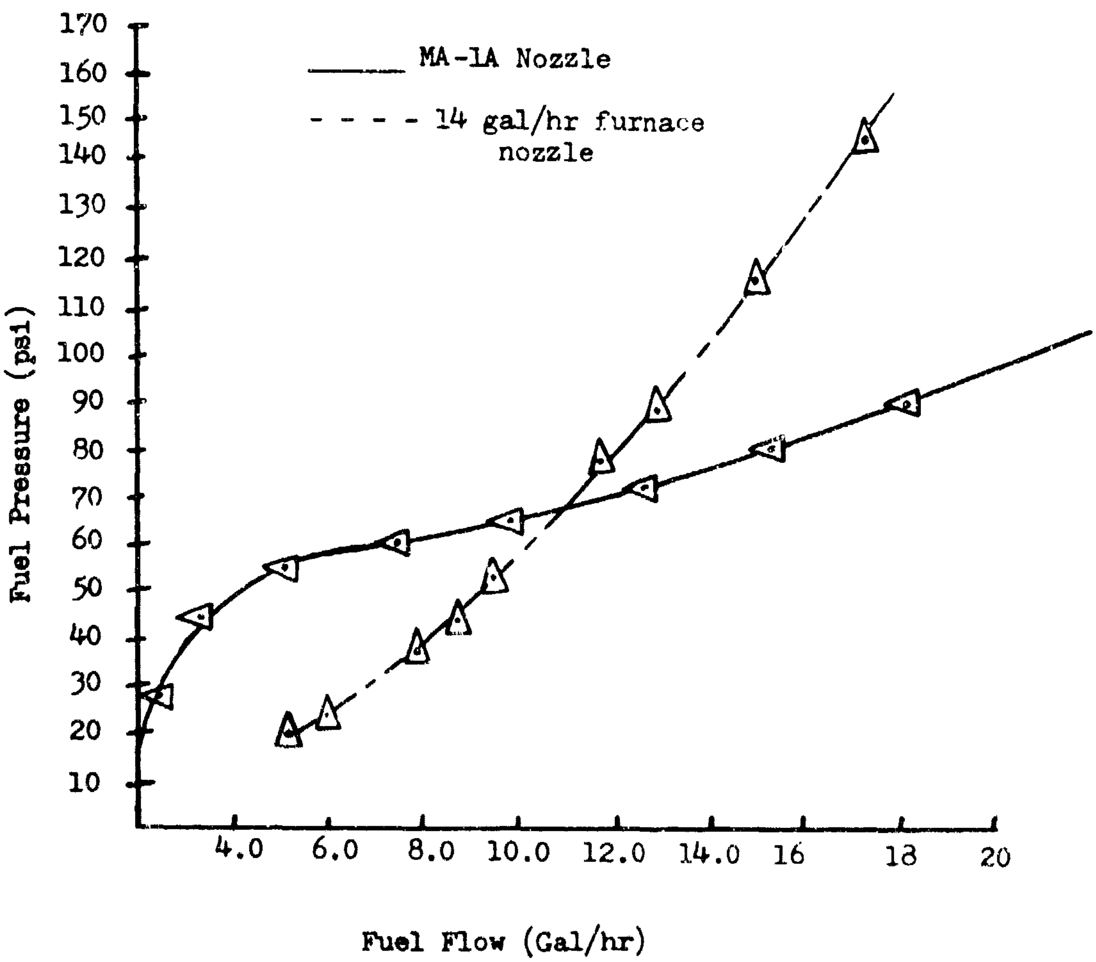


Fig. 5 Comparison of Nozzle Flow Rates

III. Burner Test Apparatus

The Air Supply

A diagram of the experimental apparatus showing Burner "A", the plenum chamber, the orifice and ducting is presented in Figure 6A. The ducting is a standard 3-inch pipe which was connected to an MA-1A JASU by a 3-inch flexible hose. Air leaving the plenum was delivered by a 2-inch pipe with a 90 degree elbow to simulate the installation on the turbocharger. The air supply was controlled by a 3-inch gate valve located 24 diameters upstream from the orifice. The air supply system was identical for the Burner "B" tests except that the 2-inch section after the plenum was replaced with a 3-inch section.

The Fuel System

A diagram of the fuel system is shown in Figure 6B. The fuel system consisted of an 80 gallon tank, a fuel filter and shut-off valve, a solenoid shut-off valve, a manual needle valve, and a flow meter. The system was pressurized to 200 psi using high pressure nitrogen.

The Ignition System

The ignition system consisted of a 10,000 volt transformer, a Variac transformer, a remote switch, an apparatus switch and the electrodes. The Variac was used to reduce voltage in order to slow down the rate of breakdown in the insulators on the locally constructed ignitors. Commercial sparkplugs did not require any special care.

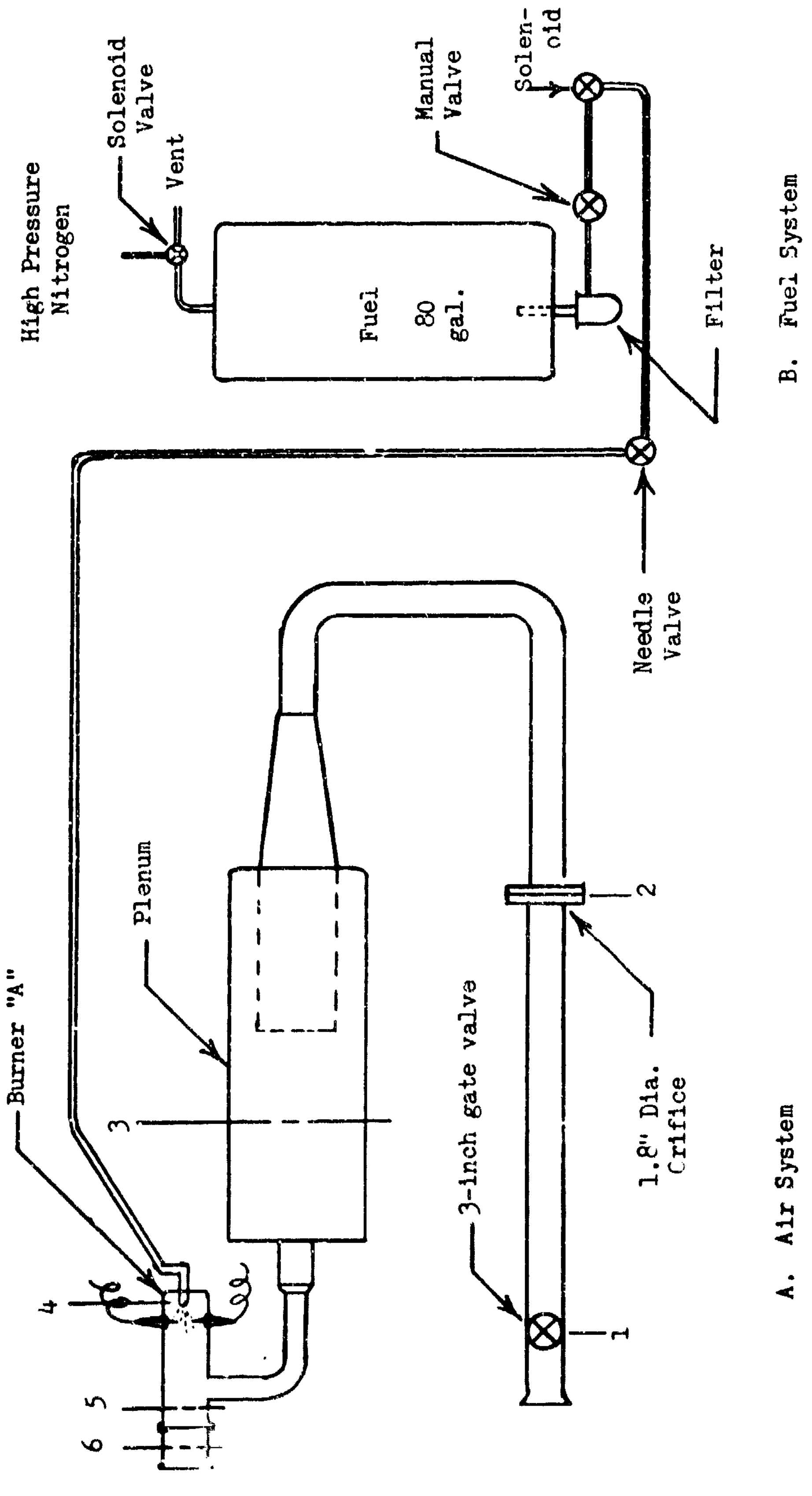


Fig. 6 Schematic of Burner Test Stand

Instrumentation

Instrumentation was located at stations 2, 3, 4, 5, and 6 as shown in Figure 6A. The following table lists the location and type of instrumentation at each station.

Table 1
Instrumentation

Station	Thermo-Couples	Static Pressure	Total Pressure
2	0	2	0
3	1	0	1
4	0	3	0
5	0	2	0
6	1	3	1

A 1.8 inch diameter square-edged orifice was installed at station 2 for metering the air flow. The thermocouple at station 6 was moveable and was used to obtain the burner exit temperature profile. This thermocouple was made of unshielded chromel-alumel wire; the one at station 3 was shielded iron-constantan. Pressure measurements were read on 5 U-tube mercury manometers, 2 precision pressure gauges calibrated in "Hg, and 3 pressure gauges reading in psi. Temperatures were recorded on 3 Honeywell temperature recorders.

IV. Analytic Development and Results

The design of the combustors was constrained by three general factors, the physical size of the turbochargers, the results of the cycle analysis from the "Design Point Turbine Engine Performance Program" (Ref. 10), and the burner areas required by the estimated mass flow. Because of the short distance between the compressor exit and the turbine inlet on the turbochargers, the combustors were designed for reverse flow. Reverse flow allows a more compact package and reduces duct lengths.

The cycle analysis showed that the temperature at the burner exit has a significant effect on engine performance, while pressure loss in the burner has a lessor effect. Figure 7 shows the effects of temperature and pressure loss variations on thrust for each turbocharger.

Calculations for determining the required areas to maintain appropriate velocities in the various burner sections were based on values obtained by using weight flow function (Ref. 3)

$$F_{(M)} = \frac{W\sqrt{T_t R}}{P_t A} \qquad (1)$$

Table 2 lists the calculated velocities in each section of Burner "A" for a flow rate of 0.75 lbm/sec. Area calculations for primary and secondary air flow were based on one dimensional flow and on chart values (Ref. 4) for minimum pressure loss and good fuel atomization. Appendix B describes the details of the calculations.

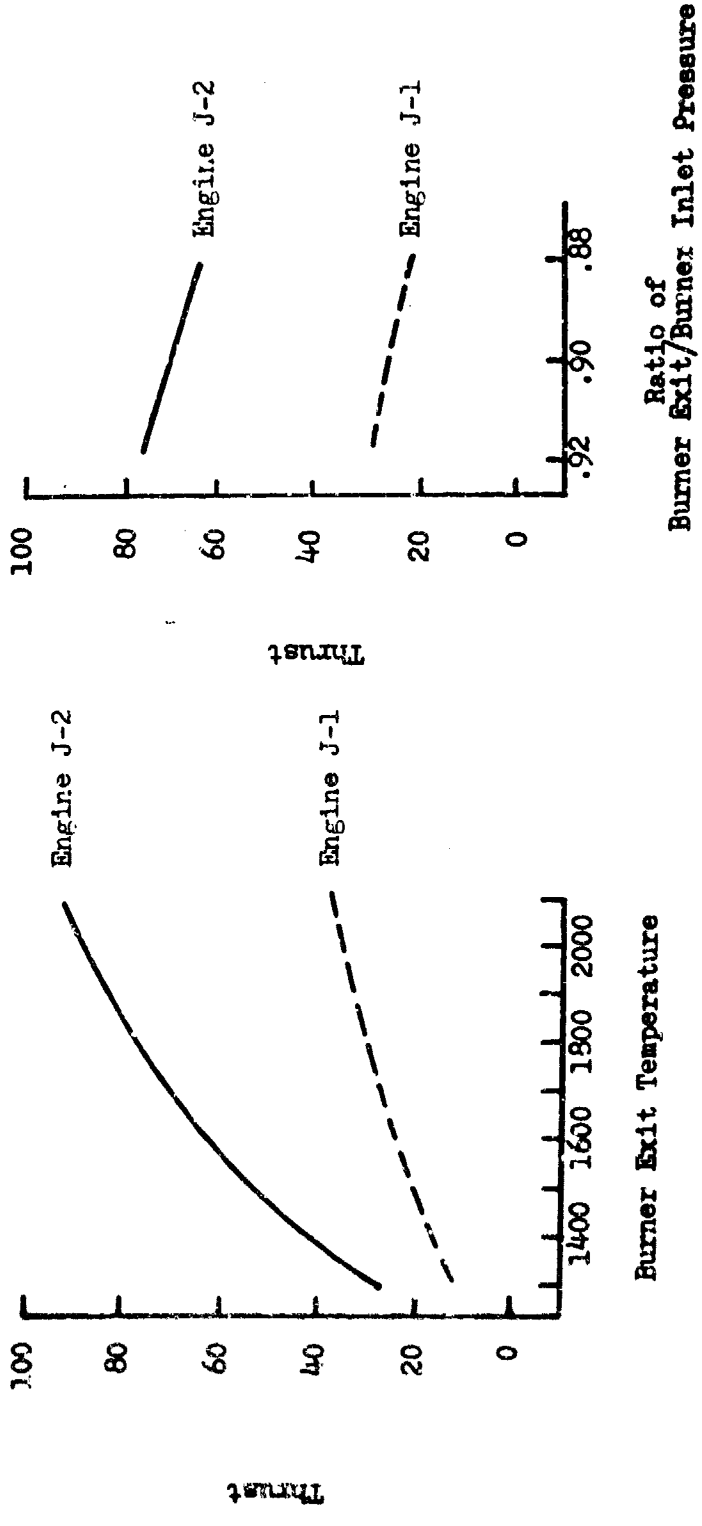


Fig. 7 Effect of Burner Exit Temperature and Pressure Drop on Engine Thrust

Burner "B" was designed in a similar manner for a flow rate of 1.5 lbm/sec except for parameters fixed by the MA-1A liner. The inlet air was injected tangentially in the burner plenum to create more uniform flow conditions and eliminate stagnation regions which occurred in the plenum of Burner "A". Table 3 lists the calculated velocities in the different sections of Burner "B".

Table 2 Burner "A" Velocities @ $\dot{m}$ = 0.75 lbm/sec

Station	Mach	Velocity
Inlet section	0.17	214 fps
Plenum mid-point	0.03	38 fps
Stoichiometric section	0.04	50 fps
Burner exit	0.25	546 fps
Adaptor exit	0.25	546 fps

Table 3 Burner "B" Velocities @ $\dot{m}$ = 1.5 lbm/sec

Station	Mach	Velocity
Inlet section	0.10	126 fps
Plenum mid-point	0.06	76 fps
Stoichiometric section	0.02	27 fps
Burner exit	0.15	340 fps
Adaptor exit	0.15	322 fps

V. Burner Test Stand Development and Results

Development of the burner on the test stand was controlled by two requirements. First, modifications were made to achieve satisfactory burner operation and second, measurements were made to determine the average exit temperature. This temperature was required for later analysis of the complete engine.

Burner Modifications

During testing of Burner "A" with the exhaust open to atmospheric pressure, burning was observed in the lower half of the plenum between the inlet and the ignitor (see Figure 8 below). A hot spot developed in this part of the combustor both on the plenum shell and on the liner, of sufficient intensity to melt down one ignitor and warp the liner.

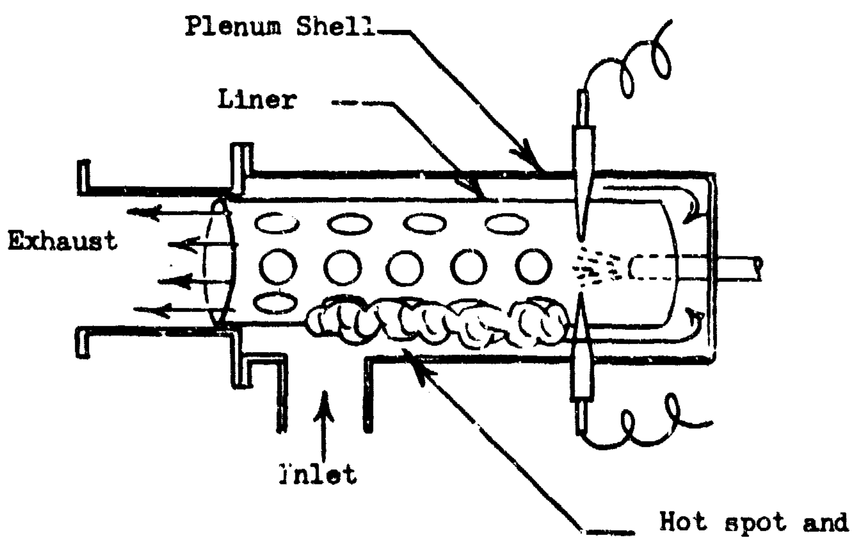


Fig. 8 - Burner "A" Hot Spot

To correct the hot spot problem, the ignitors were moved forward and a deflector vane was installed in the inlet as shown

in Figure 9. This modification cooled the hot spot enough that the liner no longer became red hot. Satisfactory operation was achieved over the expected range of the turbocharger for engine J-1.

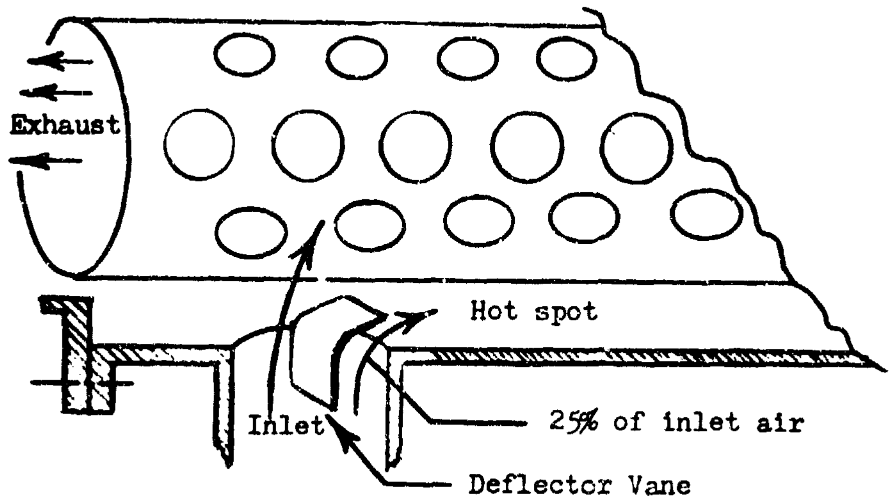


Fig. 9 Deflector Vane

Determination of Average Exit Temperature

The average exit temperature of the burner was determined by integration of the measured temperature profiles across the burner exit. These surveys were accomplished using an unshielded chromel-alumel thermocouple. The burner exit was open to atmospheric pressure.

Burner "A"

Burner "A" shows a large gradient across the exit with the highest temperatures concentrated on the inlet side of the combustor. Figure 10 shows the results of this survey.

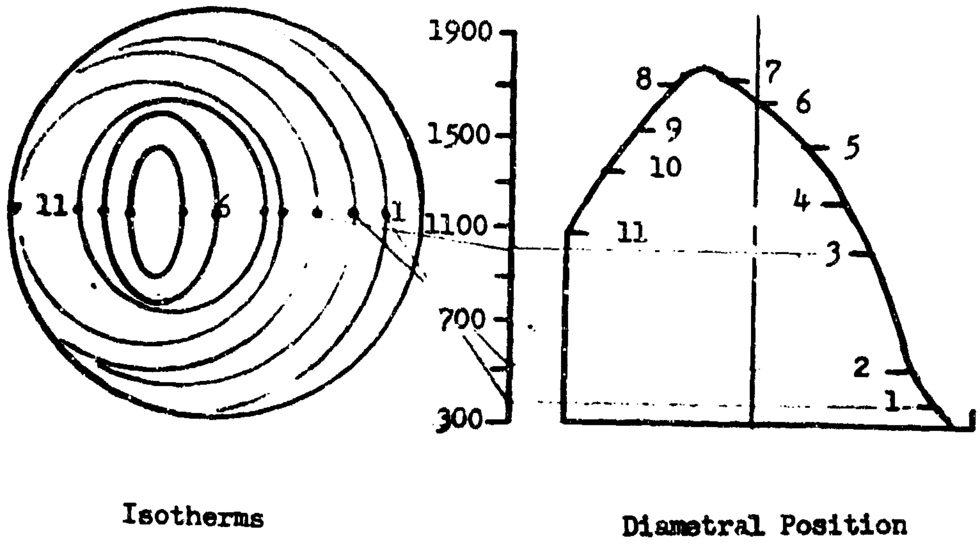
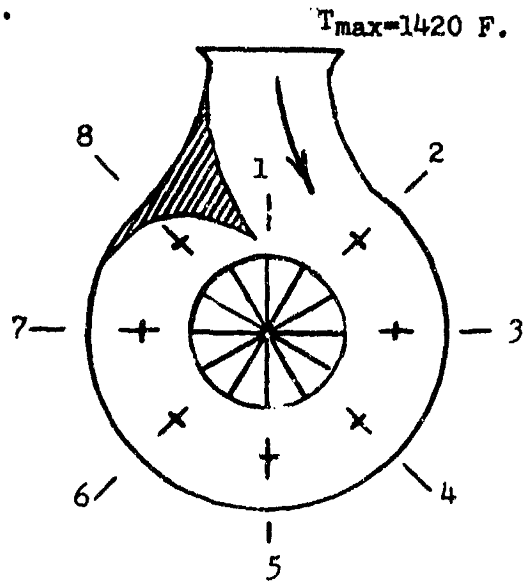


Fig. 10 - Temperature Profile Burner "A"

Attempts to reduce the exit temperature gradient by inducing swirl in the plenum, adding a swirler at the fuel nozzle, and altering secondary airflow near the burner exit were unsuccessful. Integration of the temperature profile, however, resulted in an average temperature which was 87 percent of the peak temperature.

The temperature survey in the flow channel of the turbine housing was conducted by moving the temperature axially across the channel at the points shown in Figure 11. From Point 2 to Point 4, the temperature was constant across the channel and was approximately equal to the integrated value for the average burner exit temperature. Because the turbine wheel was removed, the temperatures at Points 5 through 8 were not considered accurate due to the flow exiting through the turbine exhaust.

The temperature profiles at each point did remain flat, however, indicating that turbulent mixing in the turbine housing tended to eliminate uneven temperature distributions from the burner exit.



Point	Temperature
1	- -
2	1230°
3	1280°
4	1240°
5	1150°
6	- -
7	1050°
8	1060°

Fig. 11 - Temperature Profile in the Turbine Housing

A confirmation of the average temperature value was attempted by hot and cold flow comparisons. Using Equation 1, the relationship between mass flow at the burner exit is

$$\left[\frac{F_{(M)}^2 P_t^2}{T_t\, W^2}\right]_{Cold} = \left[\frac{F_{(M)}^2 P_t^2}{T_t\, W^2}\right]_{Hot} \tag{2}$$

assuming $\frac{\sqrt{R}}{A}$ is the same in the two cases. Choked flow was established by using a baffle plate across the burner exit and maintaining the combustion chamber pressure between 2.8 and 3.3 atmospheres. Using $\gamma = 1.4$ cold and 1.32 hot, the value for the function of Mach number (Ref. 3) was

$$F_{(M)_{Cold}} = 3.88 \qquad F_{(M)_{Hot}} = 3.79$$

Therefore, the equation

$$T_{\substack{\text{Exit}\\(\text{Hot})}} = T_{(\text{Cold})}\left(\frac{W_{(\text{Cold})}}{W_{(\text{Hot})}}\right)^2\left(\frac{P_{t(\text{Hot})}}{P_{t(\text{Cold})}}\right)^2\left(\frac{3.79}{3.88}\right)^2 \qquad (3)$$

was solved for average exit temperature.

Exit temperatures obtained by this method ranged from 80 percent to 100 percent of the measured peak temperature. This method was inconclusive because the squaring of errors in mass flow and pressure measurements could induce errors up to 20 percent in the calculated exit temperature.

Burner "B"

In view of the experience with Burner "A", Burner "B" was designed with a tangential air inlet to provide swirl of sufficient strength to develop more uniform conditions in the plenum. Test stand results showed no hot spots and a more even temperature profile. Maximum temperature was recorded on the burner centerline and showed a temperature variation of 300 F. from the liner wall to the center. Testing on the turbocharger, however, resulted in unstable operation even though the burner alone was stable. During further testing, it was discovered that unburned gases reaching the tailpipe were afterburning intermittantly. These intermittant pulses were transmitted throughout the engine and caused severe surging. Better atomization of the fuel was effective in eliminating this problem. The furnace nozzle was replaced with a hollow cone MA-1A nozzle which began proper spray patterns at 10 psi above combustion

chamber pressure and developed a finer spray.

VI. Burner Development and Results on the Engine

Both turbocharger engines were run using their respective combustors. Results obtained fit into two categories; those having a direct effect on engine thrust and performance, and those affecting the burner alone.

Engine Performance

The small engine (J-1) operated successfully over a wide range of temperatures and pressures. Burner stability on the engine was good and no significant problems were observed. Thrust measurements were not obtained due to bearing failure in the turbocharger.

The larger turbocharger operated successfully with two different turbine housings, and achieved a thrust of 53 pounds and 55 pounds (See Ref. 5 for detailed engine performance).

Surging and instabilities prevented operation on three other turbine housings. Stability of the burner on the combustor test stand did not insure stability of the burner/turbocharger combination. A significant factor adversely affecting stability with engine operation was the presence of unburned fuel downstream of the burner exit. This unburned fuel caused intermittant afterburning which in turn caused turbocharger surging.

Intermittant afterburning and burning in the turbine occurred with even small amounts of unburned fuel present. The burning in the turbine caused significantly increased turbine temperatures.

These temperatures were not measured, but were judged qualitatively by visual observation. The turbine wheel operated without any apparent redness when downstream fuel was absent, but was bright red when burning occurred in the turbine housing. With this afterburning, the turbine exit temperature exceeded 1600 F. which was considered unacceptable for this investigation.

Non-uniform temperature distribution from the burner was reduced by turbulent mixing of the flow in the inlet duct of the turbine housing. The flow mixing of the turbine housing created relatively uniform temperatures at the turbine as shown in Figure 11. Average turbine entry temperature can be read directly by locating a probe at the end of the inlet duct (Point 3 in Fig. 11).

Burner Performance

Self-sustaining combustion in Burner "A" was not possible without the plenum burning outside the liner and the associated liner hot spots (See Fig. 8). Velocity in the combustion section of Burner "A", up to 50 ft/sec, was too high. Sustained combustion was probably achieved because the liner acted as a hot surface re-ignition source (Ref. 9). The hot surface was the result of burning outside the liner which was caused by recirculation from the inside of the liner back to the plenum.

Burner "A" had too low a velocity through the liner side holes. The turbulent velocity components inside the liner exceeded the inflow velocity causing back flow from the liner

region to the plenum. This problem was caused by too small a pressure differential across the liner, a condition shown by the burner pressure loss measurements. Burner "A" had only a 1 percent pressure loss through the entire burner during cold flow measurements.

With increased blockage to increase the pressure drop across the liner, sustained combustion was precluded. With the flow blockage in place, varying the mass flow did not produce a stable operating point. Low mass flow resulted in the flame advancing upstream and "over-rich" flame-out occurring. Increasing the mass flow resulted in "lean" flame-out. Because no constant velocity section existed, the flame front was always unstable and tended to move either upstream or downstream. As the flame front moved upstream, the velocity of the air flow decreased allowing the flame to move farther upstream into an overly rich region; as the flame moved downstream, the velocity of the air increased causing the flame to move farther downstream into an overly lean region.

VII. Conclusions and Recommendations

The concept of converting a turbocharger into a low-cost jet engine was demonstrated. Specific conclusions relating the particular equipment used in this study are listed below.

(1). Two combustors were designed and constructed which performed successfully on their respective engines. Engine J-2 achieved a maximum thrust of 55 pounds.

(2). Furnace nozzles were unsatisfactory for operation of Burner "B" due to poor atomization.

(3). Turbulent mixing in the turbine housing tended to eliminate uneven burner exit temperatures.

(4). Unburned fuel leaving the combustor caused stability problems and turbine temperature problems.

Recommendations

In order to extend the operating range of the present configuration or develop a more advanced combustor, a further investigation of the stability problem is necessary. The following studies would be valuable.

(1). A study to isolate burner instabilities from turbocharger instabilities. This study can use the present experimental apparatus by using the MA-1A JASU to drive the burner and turbine. The engine compressor air can then be discharged against a back pressure allowing individual measurements of turbine and compressor effects.

(2). A study to determine velocity profiles within the

combustor both upstream and downstream, required for stable operation.

(3). A study extending the operating range of the present J-2 engine configuration by fine-tuning measures, such as hardening up the fuel system, fuel pre-heat, and other alterations to improve combustor efficiency.

(4). A study to examine operating characteristics of the J-1 engine. Experimental equipment is available to complete this project except for the engine exhaust nozzles.

Bibliography

1. Ames Research Staff, Equations, Tables and Charts for Compressible Flow, Report No. 1135, Washington, D.C., U.S. Government Printing Office, 1953.

2. A.S.M.E., Fluid Meters, Their Theory and Application, American Society of Mechanical Engineers, New York, 1959.

3. Emelfelt, Ivan H., Graphical Presentation of Compressible Flow Functions, Technical Information Series No. R52GL-300, General Electric Technical Data Center, Schenectady, New York, 1952.

4. Hill, Philip G. & Peterson, Carl R., Mechanics and Thermodynamics of Propulsion, Reading, Massachusetts, Addison-Wesley Publishing Company, 1970, (p. 150).

5. Kent, Tommy J., Conceptual Study for the Development of Low-Cost Turbojets, Thesis, Air Force Institute of Technology, Wright-Patterson AFB, Ohio, March, 1976. Unpub.

6. MacInnes, Hugh, How to Select and Install Turbochargers, Tucson, Arizona, H. P. Books, 1971, (p. 151).

7. Taylor, John W. R., Janes All the World's Aircraft, London, 1974 - 1975.

8. Vincent, E. T., The Theory and Design of Gas Turbine and Jet Engines, New York, McGraw - Hill Book Co., Inc., 1950.

9. Weiss, Kenneth E., The Design and Performance of a Streamlined Flame Holder, Thesis, Air Force Institute of Technology, Wright-Patterson AFB, Ohio, August, 1956.

10. Witherell, R. E., Design Point Turbine Engine Performance Program, Air Force Aero Propulsion Laboratory, Technical Report No. TR-68-88, Wright-Patterson AFB, Ohio, September 1968.

Appendix A

Development of Orifice Curves

Development of Orifice Curves: (Ref. 2)

$$\dot{m}_a = 0.525\, K Y_1 d^2 \sqrt{\gamma_1 (P_1 - P_2}$$

$$R_e = \frac{4\dot{m}}{\pi D \mu} = 316{,}000\, \dot{m}$$

$$R_e \text{ range } \begin{array}{l} 158{,}000\ (\dot{m} = 0.5) \\ 475{,}000\ (\dot{m} = 1.5) \end{array}$$

$$K = \begin{array}{l} 0.6532 \text{ for } 100{,}000\ R_e \\ 0.6506 \text{ for } 600{,}000\ R_e \end{array}$$

$$K_{average} \doteq 0.652$$

$$Y_1 = 1 - (0.41 + 0.35\beta^4)\frac{x}{k} \;;\; k = 1.4$$

β chosen equal to 0.6 Dia.

Dia. of pipe 3.156" ; Dia of Orifice 1.894"

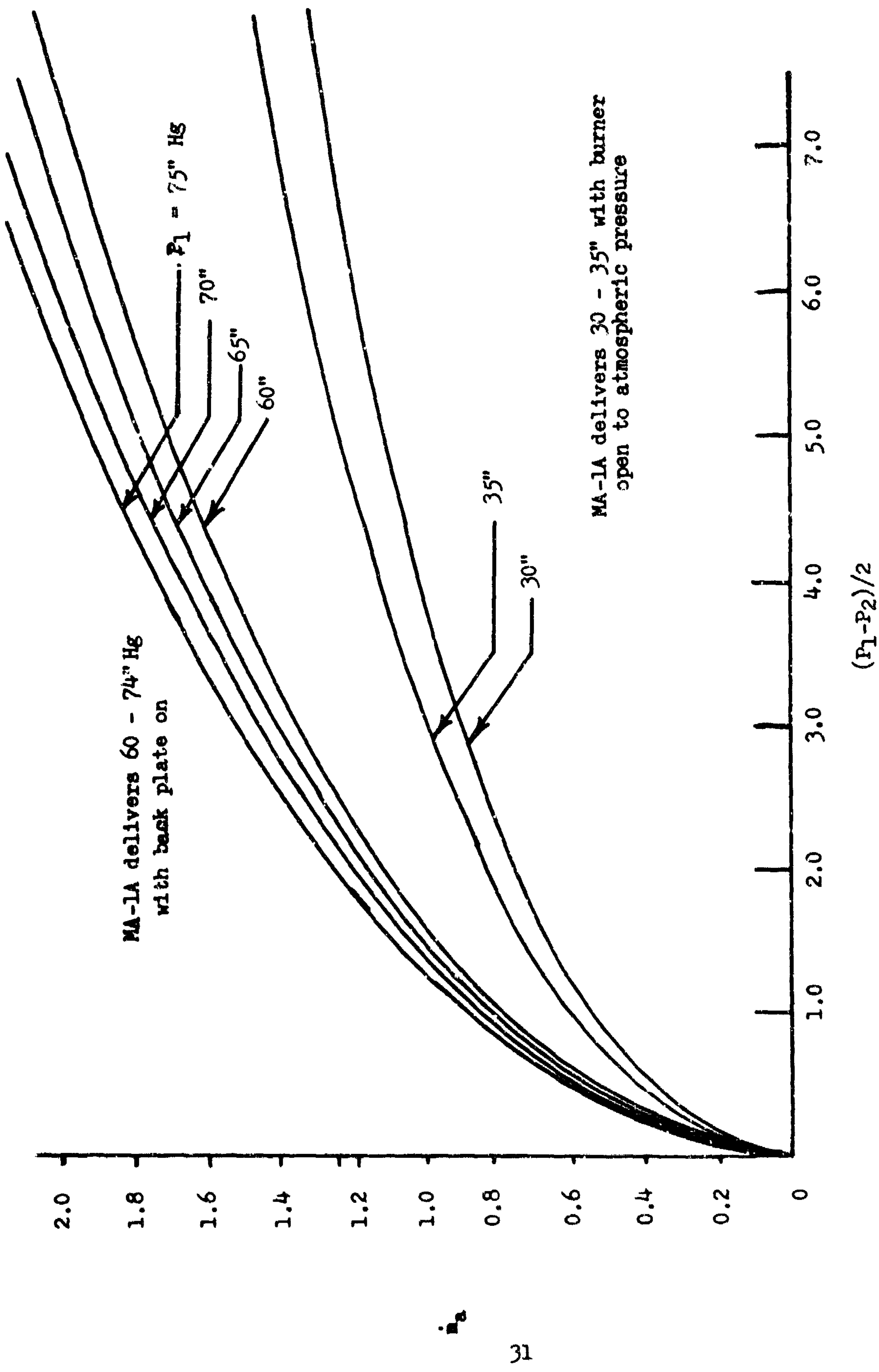


Fig. 12 Orifice Calibration Curve

Appendix B

Combustor Air Flow Calculations

Areas

Calculations for determining the required areas within the combustor were made using the weight flow function with an assigned Mach number. For calculation of the required Mach the following values were used

$$\gamma = 1.3 \quad ; \quad \gamma = 1.4$$

$$T_T = T_S$$

$$T_T = 660\ R \quad ; \quad T_T = 2160\ R$$

$$M = u/\sqrt{\gamma R T}$$

Using the weight flow function

$$F_{(M)} = \frac{W\sqrt{R\,T_T}}{P_T\,A}$$

and assuming equal distribution of mass flow throughout the burner, areas were calculated for the inlet, plenum and liner cross-section.

Primary and secondary air holes were selected for 20% at the front of the liner for fuel atomization, 13% to complete the stoichiometric section and 67% evenly distributed for cooling (Ref. 8). The total area for primary and secondary air was selected from chart values (Ref. 4) to minimize total pressure loss. The measured pressure loss in Burner "A" was 3 percent and in Burner "B" it was 8 percent.

Appendix C

Results of Cycle Analysis

using

"Design Point Turbine Engine Performance Program"

TURBOCHARGER BASELINE ENGINE (RAJAY MODEL 370 E)
BASE POINT

HPEXT =	0.00	LCSL =	0.00000	HCSL =	0.00000
PCBLOW=	0.00000	PCBLCB=	0.00000	PCBLNZ=	0.00000
PCBLC =	0.00000	PCBLT4=	0.00000	PCPLHP=	0.00000
PROA =	2.800	BPR =	0.000	WAENG =	.66

PRFT =	1.0000	ETAFT =	1.00000	WA24 =	0.000
PRFH =	1.0000	ETAFH =	1.00000	WA21 =	.663
PRLP =	1.0000	ETALP =	1.00000	WA22 =	.663
PRC =	2.8000	ETAC =	.65000	WA3 =	.663
DPD =	0.00000	DPFH =	0.00000	DPC =	0.00000

PRESSURES IN ATMOSPHERES

P1 =	1.000	T1 =	518.67	H1 =	123.92
P2 =	1.000	T2 =	518.67	H2 =	123.92
P21H =	1.000	T21 =	518.67	H21 =	123.92
P21L =	1.000				
P22L =	1.000	T22 =	518.67	H22 =	123.92
P22 =	1.000				
P3 =	2.800	T3 =	790.77	H3 =	189.46
P4 =	2.660	T4 =	2160.00	H4 =	564.92
DPB =	.0500	ETAB =	.90000	FAR4 =	.02302
P41 =	2.660	T41 =	2159.99	H41 =	564.92
P5 =	1.290	T5 =	1938.74	H5 =	500.86
PRTH =	2.0616	ETATH =	.65000	DPT =	0.00000
P51 =	1.290	T51 =	1938.74	H51 =	500.86
P55 =	1.290	T55 =	1938.74	H55 =	500.86
PRTL =	.9998	ETATL =	1.00000	DPE =	0.00000
P56 =	1.290	T56 =	1938.74	H56 =	500.86
P6 =	1.290	T6 =	1938.74	H6 =	500.86
P7 =	1.290	T7 =	1938.74	H7 =	500.86
P8 =	1.290	T8 =	1938.74	H8 =	500.86
V9 =	1285.76	TS9 =	1822.98	HS9 =	467.83

P24 =	0.000	T24 =	0.00	H24 =	0.00
P26 =	0.000	T26 =	0.00	H26 =	0.00
P27 =	0.000	T27 =	0.00	H27 =	0.00
P28 =	0.000	T28 =	0.00	H28 =	0.00
V29 =	0.00	TS29 =	0.00	HS29 =	0.00

FAR41 =	.02302	FAR51 =	.02302	FAR56 =	.02302
FAR7 =	.02302	FAR8 =	.02302	FAR27 =	0.00000
HV =	17380.2	HV7 =	0.0	HV27 =	0.0
CV =	1.00000	V9 =	1285.76	V29 =	0.00

FN/WA =	40.883	THG9 =	40.883	THG29 =	0.000
ETAP =	1.00000	ETATHM=	1.00000	ETAO =	1.00000
SFCU =	2.02747	FG =	27.1	FN =	27.1

TURBOCHARGER (RAJAY 370 E) ENGINE WITH COMP MAP A/B

HPEXT	=	0.00	LCSL	=	0.00000	HCSL	=	0.00000
PCBLOW	=	0.00000	PCBLOB	=	0.00000	PCBLNZ	=	0.00000
PCBLC	=	0.00000	PCBLT4	=	0.00000	PCBLHP	=	0.00000
PROA	=	2.800	BPR	=	0.000	WAENG	=	.66
PRFT	=	1.0000	ETAFT	=	1.00000	WA24	=	0.000
PRFH	=	1.0000	ETAFH	=	1.00000	WA21	=	.663
PRLP	=	1.0000	ETALP	=	1.00000	WA22	=	.663
PRC	=	2.8000	ETAC	=	.65000	WA3	=	.663
DPD	=	0.00000	DPFH	=	0.00000	DPC	=	0.00000

PRESSURES IN ATMOSPHERES

P1	=	1.000	T1	=	518.67	H1	=	123.92
P2	=	1.000	T2	=	518.67	H2	=	123.92
P21H	=	1.000	T21	=	518.67	H21	=	123.92
P21L	=	1.000						
P22L	=	1.000	T22	=	518.67	H22	=	123.92
P22	=	1.000						
P3	=	2.800	T3	=	790.77	H3	=	189.46
P4	=	2.660	T4	=	2160.00	H4	=	564.92
DPB	=	.0500	ETAB	=	.90000	FAR4	=	.02302
P41	=	2.660	T41	=	2159.99	H41	=	564.92
P5	=	1.290	T5	=	1938.74	H5	=	500.86
PRTH	=	2.0615	ETATH	=	.65000	DPT	=	0.00000
P51	=	1.290	T51	=	1938.74	H51	=	500.86
P55	=	1.290	T55	=	1938.74	H55	=	500.86
PRTL	=	.9998	ETATL	=	1.00000	DPE	=	0.00000
P56	=	1.290	T56	=	1938.74	H56	=	500.86
P6	=	1.290	T6	=	1938.74	H6	=	500.86
P7	=	1.226	T7	=	3960.00	H7	=	1187.83
* *	A/B	* *	ETABAB	=	.90000	DPBAB	=	.05000
P8	=	1.226	T8	=	3960.01	H8	=	1187.83
V9	=	1655.05	TS9	=	3799.70	HS9	=	1133.11
P24	=	0.000	T24	=	0.00	H24	=	0.00
P26	=	0.000	T26	=	0.00	H26	=	0.00
P27	=	0.000	T27	=	0.00	H27	=	0.00
P28	=	0.000	T28	=	0.00	H28	=	0.00
V29	=	0.00	TS29	=	0.00	HS29	=	0.00
FAR41	=	.02302	FAR51	=	.02302	FAR56	=	.02302
FAR7	=	.06722	FAR9	=	.06722	FAR27	=	0.00000
HV	=	17360.2	HV7	=	15818.2	HV27	=	0.0
CV	=	1.00000	V9	=	1655.05	V29	=	0.00
FN/WA	=	54.898	THG9	=	54.898	THG29	=	0.000
ETAP	=	1.00000	ETATHM	=	1.00000	ETAO	=	1.00000
SFCU	=	4.40785	FG	=	36.4	FN	=	36.4

TURBOCHARGER (AIRESEARCH T18A E) ENGINE
BASE POINT

HPEXT =	0.00	LCSL =	0.00000	HCSL =	0.00000
PCBLOW=	0.00000	PCBLOB=	0.00000	PCBLN7=	0.00000
PCBLC =	0.00000	PCBLT4=	0.00000	PCBLHP=	0.00000
PROA =	4.000	BPR =	0.000	WAENG =	1.39
PRFT =	1.0000	ETAFT =	1.00000	WA24 =	0.000
PRFH =	1.0000	ETAFH =	1.00000	WA21 =	1.389
PRLP =	1.0000	ETALP =	1.00000	WA22 =	1.389
PRC =	4.0000	ETAC =	.65000	WA3 =	1.389
DPD =	0.00000	DPFH =	0.00000	DPC =	0.00000

PRESSURES IN ATMOSPHERES

P1 =	1.000	T1 =	518.67	H1 =	123.92
P2 =	1.000	T2 =	518.67	H2 =	123.92
P21H =	1.000	T21 =	518.67	H21 =	123.92
P21L =	1.000				
P22L =	1.000	T22 =	518.67	H22 =	123.92
P22 =	1.000				
P3 =	4.000	T3 =	903.38	H3 =	217.00
P4 =	3.800	T4 =	2160.00	H4 =	563.75
DPB =	.0500	ETAB =	.90000	FAR4 =	.02126
P41 =	3.800	T41 =	2159.99	H41 =	563.75
P5 =	1.418	T5 =	1843.03	H5 =	472.61
PRTH =	2.6796	ETATH =	.70000	DPT =	0.00000
P51 =	1.418	T51 =	1843.03	H51 =	472.61
P55 =	1.418	T55 =	1843.03	H55 =	472.61
PRTL =	.9998	ETATL =	1.00000	DPE =	0.00000
P56 =	1.418	T56 =	1843.03	H56 =	472.61
P6 =	1.418	T6 =	1843.03	H6 =	472.61
P7 =	1.418	T7 =	1843.03	H7 =	472.61
P8 =	1.418	T8 =	1843.03	H8 =	472.61
V9 =	1457.97	TS9 =	1692.03	HS9 =	430.14
P24 =	0.000	T24 =	0.00	H24 =	0.00
P26 =	0.000	T26 =	0.00	H26 =	0.00
P27 =	0.000	T27 =	0.00	H27 =	0.00
P28 =	0.000	T28 =	0.00	H28 =	0.00
V29 =	0.00	TS29 =	0.00	HS29 =	0.00
FAR41 =	.02126	FAR51 =	.02126	FAR56 =	.02126
FAR7 =	.02126	FAR9 =	.02126	FAR27 =	0.00000
HV =	17360.2	HV7 =	0.0	HV27 =	0.0
CV =	1.00000	V9 =	1457.97	V29 =	0.00
FN/WA =	46.279	THG9 =	46.279	THG29 =	0.000
ETAP =	1.00000	ETATHM=	1.00000	ETAO =	1.00000
SFCU =	1.65398	FG =	64.3	FN =	64.3

TURBOCHARGER (AIRESEARCH T18A E) ENGINE WITH COMP MAP

```
HPEXT  =     0.00   LCSL   = 0.00000   HCSL   = 0.00000
PCBLOW= 0.00000     PCBLOB= 0.00000    PCBLNZ= 0.00000
PCBLC  = 0.00000    PCBLT4= 0.00000    PCBLHP= 0.00000
PROA   =    3.310   BPR    =   0.000   WAENG  =    1.25

PRFT   =  1.0000    ETAFT  = 1.00000   WA24   =   0.000
PRFH   =  1.0000    ETAFH  = 1.00000   WA21   =   1.250
PRLP   =  1.0000    ETALP  = 1.00000   WA22   =   1.250
PRC    =  3.3100    ETAC   =  .72000   WA3    =   1.250
DPD    = 0.00000    DPFH   = 0.00000   DPC    = 0.00000

PRESSURES IN ATMOSPHERES
P1     =    1.000   T1     =  518.67   H1     =  123.92
P2     =    1.000   T2     =  518.67   H2     =  123.92
P21H   =    1.000   T21    =  518.67   H21    =  123.92
P21L   =    1.000
P22L   =    1.000   T22    =  518.67   H22    =  123.92
P22    =    1.000
P3     =    3.310   T3     =  811.59   H3     =  194.53
P4     =    3.145   T4     = 2160.00   H4     =  564.71
DPB    =    .0500   ETAB   =  .90000   FAR4   =  .02270
P41    =    3.145   T41    = 2159.99   H41    =  564.71
P5     =    1.524   T5     = 1921.25   H5     =  495.66
PRTH   =   2.0628   ETATH  =  .70000   DPT    = 0.00000
P51    =    1.524   T51    = 1921.25   H51    =  495.66
P55    =    1.525   T55    = 1921.25   H55    =  495.66
PRTL   =    .9998   ETATL  = 1.00000   DPE    = 0.00000
P56    =    1.525   T56    = 1921.25   H56    =  495.66
P6     =    1.525   T6     = 1921.25   H6     =  495.66
P7     =    1.525   T7     = 1921.25   H7     =  495.66
P8     =    1.525   T8     = 1921.25   H8     =  495.66
V9     =  1629.07   TS9    = 1734.33   HS9    =  442.65

P24    =    0.000   T24    =    0.00   H24    =    0.00
P26    =    0.000   T26    =    0.00   H26    =    0.00
P27    =    0.000   T27    =    0.00   H27    =    0.00
P28    =    0.000   T28    =    0.00   H28    =    0.00
V29    =     0.00   TS29   =    0.00   HS29   =    0.00

FAR41  =   .02270   FAR51  =  .02270   FAR55  =  .02270
FAR7   =   .02270   FAR8   =  .02270   FAR27  = 0.00000
HV     =  17360.2   HV7    =     0.0   HV27   =     0.0
CV     = 1.00000    V9     = 1629.07   V29    =    0.00

FN/WA  =   51.783   THG9   =  51.783   THG29  =   0.000
ETAP   = 1.00000    ETATHM= 1.00000    ETAO   = 1.00000
SFCU   = 1.57816    FG     =    64.7   FN     =    64.7
```

TURBOCHARGER (AIRESEARCH T18A E) ENGINE WITH COMP MAP

HPEXT =	0.00	LCSL =	0.00000	HCSL =	0.00000
PCBLOW=	0.00000	PCBLOB=	0.00000	PCBLNZ=	0.00000
PCBLC =	0.00000	PCBLT4=	0.00000	PCBLHP=	0.00000
PROA =	4.000	BPR =	0.000	WAENG =	1.39
PRFT =	1.0000	ETAFT =	1.00000	WA24 =	0.000
PRFH =	1.0000	ETAFH =	1.00000	WA21 =	1.389
PRLP =	1.0000	ETALP =	1.00000	WA22 =	1.389
PRC =	4.0000	ETAC =	.66700	WA3 =	1.389
DPO =	0.00000	DPFH =	0.00000	DPC =	0.00000

PRESSURES IN ATMOSPHERES

P1 =	1.000	T1 =	518.67	H1 =	123.92
P2 =	1.000	T2 =	518.67	H2 =	123.92
P21H =	1.000	T21 =	518.67	H21 =	123.92
P21L =	1.000				
P22L =	1.000	T22 =	518.67	H22 =	123.92
P22 =	1.000				
P3 =	4.000	T3 =	893.73	H3 =	214.62
P4 =	3.800	T4 =	2160.00	H4 =	563.85
DPB =	.0500	ETAB =	.93000	FAR4 =	.02141
P41 =	3.800	T41 =	2159.99	H41 =	563.85
P5 =	1.459	T5 =	1851.34	H5 =	475.04
PRTH =	2.6044	ETATH =	.70000	DPT =	0.00000
P51 =	1.459	T51 =	1851.34	H51 =	475.04
P55 =	1.459	T55 =	1851.35	H55 =	475.04
PRTL =	.9999	ETATL =	1.00000	DPE =	0.00000
P56 =	1.459	T56 =	1851.35	H56 =	475.04
P6 =	1.459	T6 =	1851.35	H6 =	475.04
P7 =	1.459	T7 =	1851.35	H7 =	475.04
P8 =	1.459	T8 =	1851.35	H8 =	475.04
V9 =	1517.05	TS9 =	1687.94	HS9 =	429.07
P24 =	0.000	T24 =	0.00	H24 =	0.00
P26 =	0.000	T26 =	0.00	H26 =	0.00
P27 =	0.000	T27 =	0.00	H27 =	0.00
P28 =	0.000	T28 =	0.00	H28 =	0.00
V29 =	0.00	TS29 =	0.00	HS29 =	0.00
FAR41 =	.02141	FAR51 =	.02141	FAR56 =	.02141
FAR7 =	.02141	FAR9 =	.02141	FAR27 =	0.00000
HV =	17360.2	HV7 =	0.0	HV27 =	0.0
CV =	1.00000	V9 =	1517.05	V29 =	0.00
FN/WA =	48.161	THG9 =	48.161	THG29 =	0.000
ETAP =	1.00000	ETATHM=	1.00000	ETAO =	1.00000
SFCU =	1.60069	FG =	66.9	FN =	66.9

TURBOCHARGER (AIRESEARCH T18A E) ENGINE WITH COMP MAP A/B

```
HPEXT  =     0.00   LCSL   = 0.00000   HCSL   = 0.00000
PCBLOW= 0.00000     PCBLOB= 0.00000    PCBLNZ= 0.00000
PCBLC  = 0.00000    PCBLT4= 0.00000    PCBLHP= 0.00000
PROA   =    3.310   BPR    =   0.000   WAENG  =    1.25

PRFT   =   1.0000   ETAFT  = 1.00000   WA24   =   0.000
PRFH   =   1.0000   ETAFH  = 1.00000   WA21   =   1.250
PRLP   =   1.0000   ETALP  = 1.00000   WA22   =   1.250
PRO    =   3.3100   ETAC   =  .72000   WA3    =   1.250
DPD    = 0.00000    DPFH   = 0.00000   DPC    = 0.00000

PRESSURES IN ATMOSPHERES
P1     =    1.000   T1     =  518.67   H1     =  123.92
P2     =    1.000   T2     =  518.67   H2     =  123.92
P21H   =    1.000   T21    =  518.67   H21    =  123.92
P21L   =    1.000
P22L   =    1.000   T22    =  518.67   H22    =  123.92
P22    =    1.000
P3     =    3.310   T3     =  811.59   H3     =  194.53
P4     =    3.145   T4     = 2160.00   H4     =  564.71
DPB    =    .0500   ETAB   =  .93000   FAR4   =  .02270
P41    =    3.145   T41    = 2159.99   H41    =  564.71
P5     =    1.524   T5     = 1921.25   H5     =  495.66
PRTH   =   2.0628   ETATH  =  .70000   DPT    = 0.00000
P51    =    1.524   T51    = 1921.25   H51    =  495.66
P55    =    1.525   T55    = 1921.25   H55    =  495.66
PRTL   =    .9998   ETATL  = 1.00000   DPF    = 0.00000
P56    =    1.525   T56    = 1921.25   H56    =  495.66
P6     =    1.525   T6     = 1921.25   H6     =  495.66
P7     =    1.448   T7     = 3960.00   H7     = 1187.82
 * *   A/B   * *    ETABAB=   .93000   DPBAB  =  .05000
P8     =    1.448   T8     = 3960.01   H8     = 1187.82
V9     = 2212.75    TS9    = 3672.99   HS9    = 1090.01

P24    =    0.000   T24    =    0.00   H24    =    0.00
P26    =    0.000   T26    =    0.00   H26    =    0.00
P27    =    0.000   T27    =    0.00   H27    =    0.00
P28    =    0.000   T28    =    0.00   H28    =    0.00
V29    =     0.00   TS29   =    0.00   HS29   =    0.00

FAR41  =   .02270   FAR51  =  .02270   FAR56  =  .02270
FAR7   =   .06721   FAR8   =  .06721   FAR27  = 0.00000
HV     = 17360.2    HV7    = 15818.2   HV27   =     0.0
CV     = 1.00000    V9     = 2212.75   V29    =    0.00

FN/WA  =   73.397   THG9      73.397   THG29  =   0.000
ETAP   = 1.00000    ETAT    1.00000    ETAO   = 1.00000
SFCU   = 3.29667    FG     =    91.7   FN     =    91.7
```

```
TURBOCHARGER (AIRESEARCH T18A E) ENGINE WITH COMP MAP A/B

   HPEXT  =     0.00   LCSL   = 0.00000   HCSL   = 0.00000
   PCBLOW= 0.00000   PCBLOB= 0.00000   PCBLNZ= 0.00000
   PCBLC  = 0.00000   PCBLT4= 0.00000   PCBLHP= 0.00000
   PROA   =    4.000   BPR    =    0.000   WAENG  =    1.39

   PRFT   =   1.0000   ETAFT  = 1.00000   WA24   =   0.000
   PRFH   =   1.0000   ETAFH  = 1.00000   WA21   =   1.389
   PRLP   =   1.0000   ETALP  = 1.00000   WA22   =   1.389
   PRC    =   4.0000   ETAC   =  .66700   WA3    =   1.389
   DPD    =  0.00000   DPFH   = 0.00000   DPC    = 0.00000

   PRESSURES IN ATMOSPHERES
   P1     =    1.000   T1     =  518.67   H1     =  123.92
   P2     =    1.000   T2     =  518.67   H2     =  123.92
   P21H   =    1.000   T21    =  518.67   H21    =  123.92
   P21L   =    1.000
   P22L   =    1.000   T22    =  518.67   H22    =  123.92
   P22    =    1.000
   P3     =    4.000   T3     =  893.73   H3     =  214.62
   P4     =    3.800   T4     = 2160.00   H4     =  563.85
   DPB    =    .0500   ETAB   =  .90000   FAR4   =  .02141
   P41    =    3.800   T41    = 2159.99   H41    =  563.85
   P5     =    1.459   T5     = 1851.34   H5     =  475.04
   PRTH   =   2.6044   ETATH  =  .70000   DPT    = 0.00000
   P51    =    1.459   T51    = 1851.34   H51    =  475.04
   P55    =    1.459   T55    = 1851.35   H55    =  475.04
   PRTL   =    .9998   ETATL  = 1.00000   DPE    = 0.00000
   P56    =    1.459   T56    = 1851.35   H56    =  475.04
   P6     =    1.459   T6     = 1851.35   H6     =  475.04
   P7     =    1.386   T7     = 3960.00   H7     = 1187.79
    * *  A/B  * *      ETABAB=  .90000   DPBAB  =  .05000
   P8     =    1.386   T8     = 3960.01   H8     = 1187.79
   V9     =  2082.49   TS9    = 3705.89   HS9    = 1101.16

   P24    =    0.000   T24    =    0.00   H24    =    0.00
   P26    =    0.000   T26    =    0.00   H26    =    0.00
   P27    =    0.000   T27    =    0.00   H27    =    0.00
   P28    =    0.000   T28    =    0.00   H28    =    0.00
   V29    =     0.00   TS29   =    0.00   HS29   =    0.00

   FAR41  =   .02141   FAR51  =  .02141   FAR56  =  .02141
   FAR7   =   .06719   FAR8   =  .06719   FAR27  = 0.00000
   HV     =  17360.2   HV7    = 15818.2   HV27   =     0.0
   CV     =  1.00000   V9     = 2082.49   V29    =    0.00

   FN/WA  =   69.075   THG9   =  69.075   THG29  =   0.000
   ETAP   =  1.00000   ETATHM= 1.00000   ETAO   = 1.00000
   SFCU   =  3.53191   FG     =    95.9   FN     =    95.9
```

Vita

Raymond Lawrence Greene was born on 19 February 1939 in Detroit, Michigan, the son of Robert E. and Natalie I. Greene. He graduated in June 1957 from Taylor Center High School, Taylor Center, Michigan and attended the United States Air Force Academy graduating in 1964 with a Bachelor of Science degree in Engineering. After graduating, he was commissioned and assigned to Moody Air Force Base, Georgia for undergraduate pilot training. From Moody, he was assigned to Reese Air Force Base, Texas as a T-38 instructor. In January 1969, he was reassigned to MacDill Air Force Base, Florida for F-4 upgrading enroute to Southeast Asia. He was assigned to Ubon Royal Thai Air Force Base from September 1969 until July 1970. From September 1970 until June 1974 he was assigned to Bentwaters Royal Air Force Base, England serving various positions in operations and maintenance. After completion of this assignment he was selected to attend the Air Force Institute of Technology in the graduate Aeronautical Engineering Program.

Permanent Address: Route 2, Box 250-B
Lake Wales, Florida

This thesis was typed by: Mrs. Donna V. Greene